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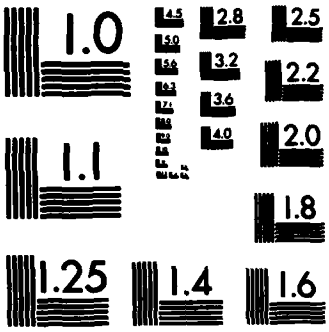

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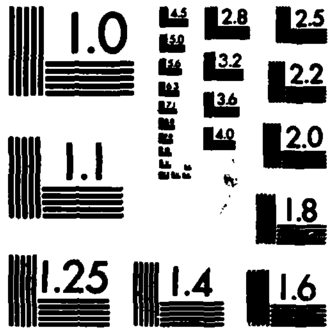

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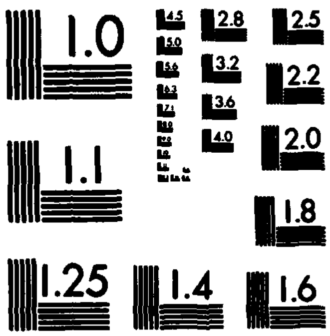

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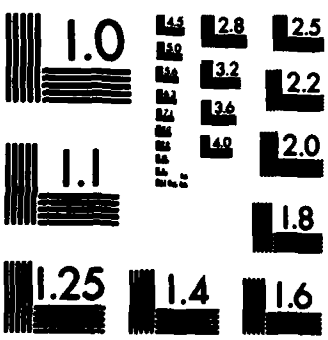

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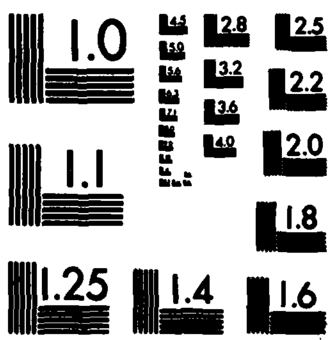

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FTD-ID(RS)T-1101-82

FOREIGN TECHNOLOGY DIVISION

AIR FORCE SYSTEMS COMMAND

A STUDY OF THE ELECTRIC ROCKET (ION ENGINE)

by

Wang Nanhao

FOREIGN TECHNOLOGY DIVISION

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A Study of the Electric Rocket (Ion Engine)

by Wang Nanhao

March, 1982

I. Survey

II. Special Features in the Study of the Electric Rocket (Ion Engine)

III. Conceptions and Views

A Study of the Electric Rocket (Ion Engine)

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I. Survey

The electric rocket is also called the electric thruster or electric propulsion (some nations call it the ion engine). It is a type of rocket engine with high specific impulses, low thrust and long life.

The building of electric rocket engine thrust is the same as any other type of propulsion engine (liquid-propellant rocket engines, air propulsion engines, nuclear propulsion engines etc.), it depends on the acceleration and jet of the working substance, the method of the acceleration or jet of the working substance or use of electric heating, or the method of using electric and magnetic volume force. Therefore, based on the definition of the electric rocket, we can divide it into three basic categories:

1. The electrothermal rocket propels the gas by electric heating and afterwards it expands to produce thrust in a suitable nozzle area.

2. The electrostatic rocket (ion engine) produces thrust by means of propellant particles (ions) with added electrostatic field accelerated jet charge.

3. The electromagnetic rocket propels the ionized propellant

flow by means of the mutual action of the external magnetic field and internal magnetic field, and the electric current which causes it to accelerate and produce thrust.

The common features of the electric rocket engines are:

1. The electric rocket engine is an engine which separates the energy source and working substance. Because of this, its economy is not like that of thermochemical propulsion engines (liquid-propulsion engines, solid-propulsion engines and gas-propulsion engines) which are only determined by one parameter - the specific impulse. It is determined by two independent parameters - specific impulse and efficiency (i.e., the efficiency of the input electric power being transformed into thrust).

Specific impulse

$$I_s = \frac{T}{\dot{M}_p g_0} \qquad (1)$$

Efficiency

$$\eta_e = \frac{P_B}{P_t} = \frac{\frac{1}{2}\dot{M}_p U_b^2}{P_t} = \frac{\text{流束的有效动能}\ (1)}{\text{输入的电功率}\ (2)} \qquad (2)$$

Key: 1. Effective kinetic energy of flux
2. Electric power of input

2. It is always hoped, as far as possible, to have the largest efficiency value under any conditions. At the same time,

under certain conditions, the specific impulse possesses the optimum value (corresponding to the minimum quantity of the energy source system). This optimum value is related to the original data of certain energy source equipment and engine equipment. It is also related to determining a spacecraft's preset maneuvering parameters under the action of electric rocket thrust.

3. In principle, it is possible to attain specific impulse for an electric rocket engine many times greater than for a liquid-propellant rocket engine because of the high concentration provided by the energy of the working substance. Theoretically, the concentration of the working substances' energy and the velocity of the jet are not limited (because it is theoretically possible for the electric energy to transform without loss into mechanical energy - kinetic energy).

Actually, it is natural in electric rocket engines that joule losses and flow losses occur in heating certain structural components (in electrostatic rocket engines). The joule loss in electrostatic rocket engines mainly appear on the working substance of electric conduction (plasma). Therefore, there is an aggregation limit in supplying energy. This is due to the limitations of the heat-resistant qualities of the structural materials. However, the flow velocity of the working substance corresponding to this limitation is many times greater than that of the thermal propulsion rocket. Therefore, the specific impulse of the electromagnetic rocket engine can reach 10^4 seconds and the specific impulse of the electrostatic rocket engine can reach 10^5 seconds (loss is small, and therefore the specific impulse is high).

4. The special feature of the electric rocket engine is that it has a very large specific mass.

$$\alpha = \frac{M_W}{P_t} \frac{2\eta_e M_W}{T v_B} \quad (3)$$

The thrust of the electric rocket engine is very small and the mass of the power equipment supplying energy is very large exceeding the thrust many times. Therefore, the electric rocket can only be used in (? illegible) and when the effective load is large it can only be used in space.

As regards the categories and common features of electric rockets, based on investigations and summaries of foreign developments in electric rockets we proposed the making of electrostatic type rockets (i.e., ion engines) according to our specific circumstances. This was in light of the fact that, in principle, the ion engine has many advantages. They are mainly: they can guarantee a high specific impulse value (2,000-10^5 seconds, there is relatively high efficiency in this specific impulse range) in a high efficiency value; the temperature during operations is quite low which actually eliminates the problem of cooling the structural components; they can be repeatedly started many times (internationally, close to 10,000 times) and can operate continuously or with interruption; life-time is long (exceeding 10,000 hours); thrust is small requiring long propulsion time and they can raise control accuracy when controlling aircraft. One of the major objectives in our study of ion engines is their application in space, as well as to carry out fundamental research.

II. Special Features in the Study of the Electric Rocket (Ion Engine)

(1) Mission Analysis

Following the continual increase of design accuracy for trajectory aircraft and propulsion systems, it is necessary to carry out mission analysis to determine the occasions for use and development avenues for the electric rocket. In a sense, mission analysis is similar to forming a complex, complete system design from many related separate systems. To accomplish mission analysis, it is first necessary to resolve the actuality of the mission and finally carefully design the flight mission.

Mission analysis typically begins from parametric study of trajectory to the desired objective. In this type of study, we find the size of the power source, the weight of the aircraft, the required type of boost for the aircraft, and the general range of the flight time and specific impulse. Afterwards, the preliminary document design of the aircraft and propulsion system will show whether or not it can use present technology to complete the mission given all of these parameters. If it is satisfactory, then the detailed design of the aircraft and propulsion system can be completed so as to determine a realistic compromise between related and unrelated variables, for example, between the weight, model and number of thrusters, the solar array design and the size of the power supply and the specific impulse, launch data and flight time. Afterwards, this type of design research becomes the basis for even more detailed trajectory analysis and is used to adapt to the actual thruster number of the aircraft. Then, we will obtain a set of improved mission analysis curves from this type of trajectory analysis and from them complete the second detailed aircraft design. The second aircraft design is commonly used to accomplish the final determination of the trajectory (this can be repeated once more if necessary).

After the mission is determined, we must understand the majority of separate systems of the electric rocket so as to obtain a complete analytical model of the propulsion system. The final optimum design must, at the same time, optimize the propulsion system and trajectory.

(2) Design Calculations

Based on the object determined by mission analysis, we can first use the following formulas to estimate thrust T required by the electric rocket engine:

$$T = 0 \cdot 65 \times 10^{-2} \frac{m_s}{\Delta \tau} g \quad \text{(半连续的推进修正)} \quad (1)$$

$$T = 3 \cdot 59 \times 10^{-9} m_s g \quad \text{(连续的推进修正)} \quad (2) \qquad (4)$$

$$T = 1 \cdot 53 \times 10^{-4} \frac{m_s}{\Delta \tau} g \quad \text{(脉冲的推进修正)} \quad (3)$$

Key: 1. Semi-continuous propulsion correction
2. Continuous propulsion correction
3. Pulsed propulsion correction

We then carry out a series of calculations for the electric rocket engine:

1. Thrust

$$T = \frac{d}{d\tau}(M_p V_B) = \dot{M}_p^+ V_B \quad (1)\ \text{(牛顿)} \qquad (6)$$

Key: 1. (Newton)

*Translator's note: No formula (5) in the original

2. Specific impulse

$$I_s = \frac{T}{\dot{M}_p g} \quad (秒)_{(1)} \qquad (7)$$

Key: 1. (Seconds)

3. Utilization ratio of propellant

$$\eta_m = \frac{\dot{M}_p^+}{\dot{M}_p} \quad (\%) \qquad (8)$$

4. Beam

$$I_B = e\frac{\dot{M}_p^+}{M_p} = \frac{e\eta_m T}{M_p g I_s} \quad (安)_{(1)} \qquad (9)$$

Key: 1. Amperes

5. Beam power (output power)

$$P_B = I_B V_B = \frac{1}{2}\dot{M}_p^+ v_B^2 \quad (瓦)_{(1)} \qquad (10)$$

Key: 1. (Watts)

6. Beam velocity

$$\upsilon_B = \sqrt{\frac{2e}{M_p} V_B} \qquad (米/秒)(1) \qquad (11)$$

Key: 1. (Meters/second)

7. Thruster's power supply efficiency (see formula (2))

$$\eta_e = \frac{I_B V_B}{P_t} \qquad (\%) \qquad (12)$$

8. Discharge power

$$P_D = I_D V_D \qquad (瓦)(1) \qquad (13)$$

Key: 1. (Watts)

9. Ion produced energy

$$\varepsilon_B = \frac{I_D V_D}{I_B} \; 或(1) \; = \frac{I_D V_D + I_{DK} V_{DK}}{I_B} \qquad (eV/ion) \qquad (14)$$

Key: 1. Or

10. Thruster's total power

$$\eta_t = \eta_e \cdot \eta_m$$

$$\eta_t = \eta_e \cdot \eta_m \cdot \gamma^2 \qquad (\%) \qquad (15)$$

11. Propellant's flow (expressed in equivalent ampere)

$$= 9.649 \times 10^{7} \times \frac{\dot{M}_p}{u} \quad (安)\ (1)$$

$$= 1.547 \times 10^{[illegible]} \times \frac{G}{T} \quad (安)\ (2) \qquad (11)$$

Key: 1. Ampere
2. Ampere

12. Total input power

$$P_t = I_B V_B + I_{ac} V_{ac} + (I_B + I_D) V_D + I_{DK} V_{DK} + I_{DH} V_{DH} + I_{DV} V_{DV} + I_{NK} V_{NK} + I_{NH} V_{NH} + I_{NV} V_{NV} + 其它 \quad (瓦)\ (1) \qquad (12)$$

Key: 1. (Watts)

There are many other formulas for the design calculations of the electric rocket but we will not list each of them here.

By using the above calculations we can consider an electric rocket prototype (including a performance prototype, engineering prototype and flying prototype). Based on the demands of mission analysis, each prototype must undergo stringent ground tests so that the electric rocket will gradually be perfected and finally meet the demands for use in space.

We selected an ion engine with an anode diameter of 8 centimeters with thrust between 3-6 mN.

(3) Each Separate System of the Electric Rocket (Ion Engine) and Its Working Contents

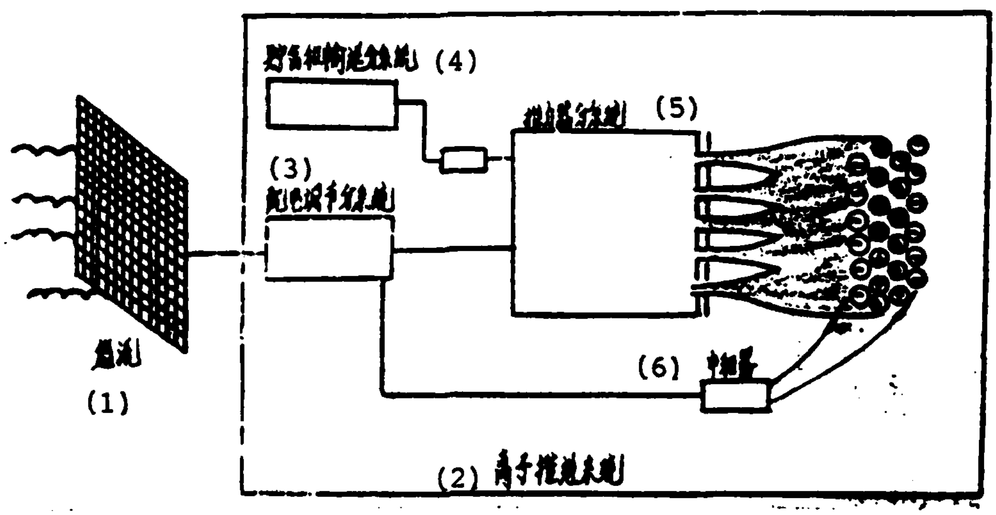

Fig. 1 Schematic of the Electric Rocket System

Key: 1. Energy source
2. Ion propulsion system
3. Distribution regulating separate system
4. Storage tank and delivery separate system
5. Thruster separate system
6. Neutralizer

The basic components of the electric rocket are the energy source and ion propulsion system (see fig. 1). The energy source should be operated by a special unit. The ion propulsion system is the major content required for the research of the ion engine. It is divided into three main separate systems: (1) thruster separate system; (2) storate tank and delivery separate system; (3) distribution regulating separate system (supplies the power source and control for the thruster separate system and storage tank and delivery separate system).

The thruster separate system includes the ion source, ion acceleration system (also called the ion extraction system) and neutralizer. Aside from the distribution regulating separate system, fig. 2 shows the structure of the electric rocket (ion engine).

(see next page for Fig. 2)

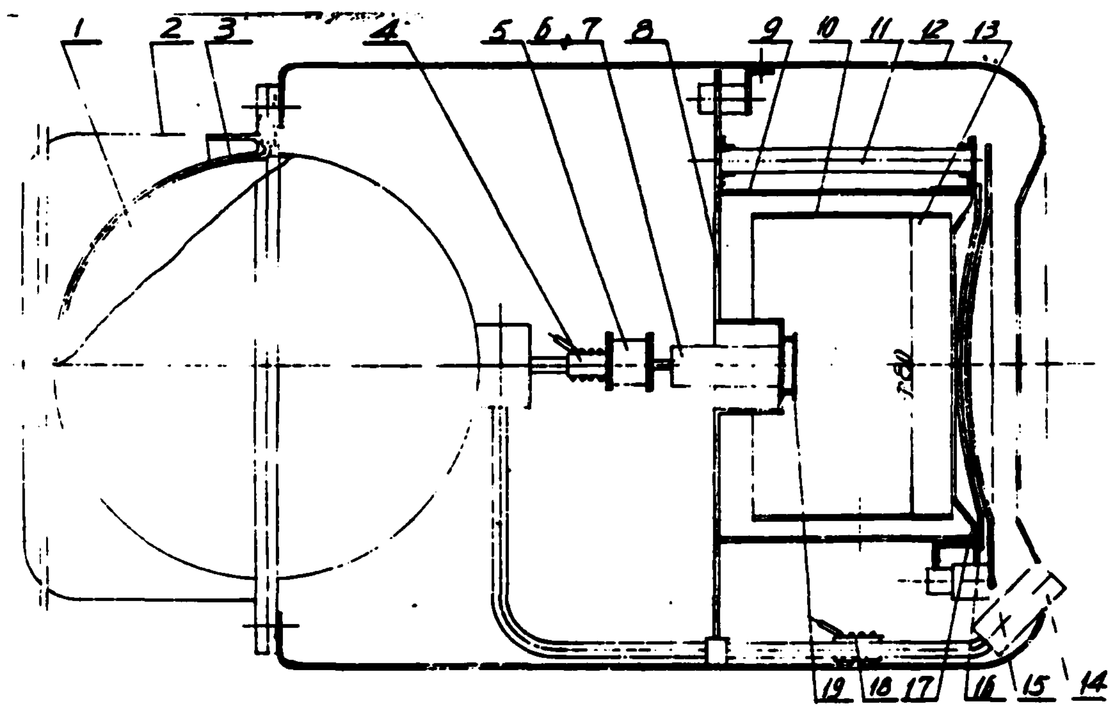

Fig. 2. Schematic of the Structure of the Ion Engine.

Fig. 2 Schematic of the Structure of the Ion Engine

Key:
1. Mercury storage tank
2. Compressor storage tank
3. (? illegible)
4. Main evaporator
5. High pressure insulator
6. Main hollow cathode
7. Circulating electrode
8. Bottom plate
9. Ion flow casing
10. Anode
11. Permanent magnet
12. Engine casing (? illegible)
13. Discharge chamber
14. Neutralizer's hollow cathode
15. Accelerating grid
16. Screen grid
17. Screen grid pole shoe
18. Neutralizer evaporator
19. Electron baffle plate

Several aspects of the operating contents of the electric rocket (ion engine) are listed below:

1. The Research and Testing of the Ion Source

The ion source is mainly composed of the cathode, anode, discharge chamber, pole shoe, magnetic rod and propellant distributor. The anode opposite the two terminal plates maintains the positive potential causing the electrons transmitted by the cathode to use the plasma layer and enter the potential region in accordance with the changes of radial and axial distances. The approximate axial magnetic field is used to prevent the electrons from directly reaching the cylindrical anode. In this type of crossover electromagnetic field structure, the electrons will spirally encircle the magnetic line of force and have longitudinal oscillation. Because of the results of the ionization by collision between the electrons and propellant gas atoms there will be a continuous supply of plasma with ions and electrons. The initial electrons will diffuse the lost energy

and line of force to the anode and be collected there resulting in the electric current going from the anode to the cathode. This type of neutral plasma fills the discharge chamber. In principle, this type of discharge mechanism can be used to cause the ionization of any gas. In reality, the most commonly used is the propellant made from (? illegible) and mercury. We used mercury to make the propellant.

Research of the ion source includes: the investigation of the physical model of the ion source's discharge chamber; investigation of the manufacture of ion sputtering of the discharge chamber; the theoretical calculation of the doubly charged ion component; computer simulation of the ion chemistry in the ion extraction system (to study the effect of beam transmission on thrust); the theoretical calculation of the charge exchange ion bombardment sputtering on the downstream surface of the accelerating electrode; understanding the manufacture of the ion beam neutralizer; and understanding the manufacture of the high vacuum insulator.

Tests of the ion source include: ion source performance tests and performance optimization tests. The aim of these tests is to design the optimum ion engine structure so that together with the distribution regulating separate system we can carry out joint tests and complete engine thermal performance tests on the ion engine. Ion source tests are a major work content in the research of ion engines and whether abroad or domestically, the focus of the work must be put into testing.

Some optimum ion sources also require a good ion extraction system. Usually, the ion drawing system comprehensively considers such factors as beam flows, beam voltage specific impulse, the minimum accelerating voltage (Vt min) required to prevent electron reverse current, the minimum total accelrated voltage (Vt min) required

for extracted beam current, the area ratio of the screen grid opening (Fos), maximum net —— total accelerating voltage ratio (Vn/Vt), the beam's divergence angle (α), the factor influencing discharge (utilization efficiency of the propellant, discharge loss, discharge voltage), the experienced transmitting environment and the repeating heat cycle. Based on the following equations (equations derived from the Charles' law) we can describe the characteristics of the ion extraction system:

Current density

$$J_B = \frac{4\varepsilon_0}{9}\left(\frac{2q}{m}\right)^{1/2} \cdot \frac{V_t^{3/2}}{l_e^2} \tag{18}$$

Current (single opening current)

$$I_B = \frac{8\varepsilon_0}{9}\left(\frac{q}{2m}\right)^{1/2} \cdot \frac{A_0}{l_e^2} \cdot V_t^{3/2} \tag{19}$$

$$I_B = \frac{\pi\varepsilon_0}{9}\left(\frac{2q}{m}\right)^{1/2} \cdot \frac{d_s^2}{l_e^2} \cdot V_t^{3/2}$$

In considering the heat distortion when the ion source is working, the two grid electrodes (the screen grid and accelerating grid) of the ion extraction system both become dish shapes. To lower the thrust loss caused by forming dish shaped grid electrodes, it is necessary to carry out distortion compensation for the dish shaped grid electrodes. The compensation values are:

$$\delta_1 = \frac{2 r_H h^2 \left(\frac{2g}{r_0^2 + h^2} + \frac{t_s + t_{ac}}{r_0^2 - h^2} \right)}{(r_0^2 + h^2) \arcsin\left(\frac{2rh}{r_0^2 + h^2} \right)} \tag{20}$$

$$\delta_2 = \delta_1 + \frac{2h d_{ac}}{(r_0^2 + h^2) K}$$

Our calculations of the ion chemistry system were carried out on a Z-80 microcomputer.

2. Research and Testing of the Evaporator

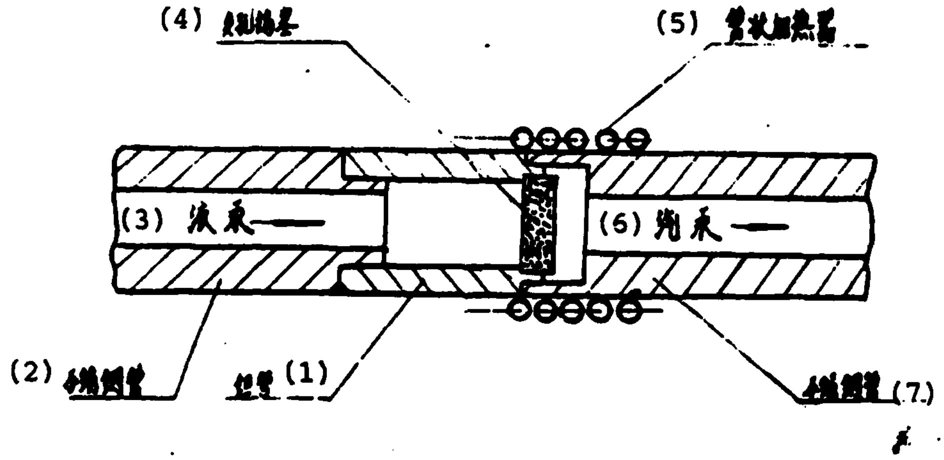

Fig. 3. Schematic of the Evaporator

Key: 1. Tantalum tube
2. Stainless steel tube
3. Liquid mercury
4. Porous tungsten plug
5. Tubular heater
6. Gas mercury
7. Stainless steel tube

Fig. 3 is a schematic of the structure of the evaporator.

It is a component in the ion engine's transmission system. Its function is to act as a steam separator of the mercury so that afterwards the liquid state mercury in the storage tank will be heated and evaporate into a steam state mercury with a certain flow rate to supply the thruster discharge chamber and hollow cathode. It can also prevent the liquid mercury from entering the discharge chamber.

Testing of the evaporator includes:

(1) Air flow tests to measure the rate of the air flow through the evaporator under different pressures.

(2) Mercury flow tests. The evaporator's measuring flow is the function of the temperature. Its flow size is controlled by the heater.

(3) Penetration tests test the limit pressure of the liquid mercury penetrating the porous tungsten (mercury and tungsten are non-penetrating and therefore when the liquid mercury comes in contact with the porous tungsten capillaries it cannot go through until the liquid mercury pressure exceeds the liquid mercury's meniscus and forms additional pressure). The osmotic pressure can reach 10 atmospheric pressure.

(4) Mercury flow tests are carried out again after the penetration tests to show that before and after porous tungsten penetration there was basically no change in the mercury flow.

(5) Life-time tests are mainly for testing the rationality of the structure, the changes of the circulation characteristics after long use and for determining a life-time of the evaporator.

The life-time of our evaporator has already reached 6,500 hours. The steam mercury flow of the evaporator has basically not changed and it can satisfy the requirements of electric

rocket tests.

3. Cathode Research

Based on mission analysis and the operational features of the electric rocket, it is necessary to manufacture two types of cathodes. One type is the discharge chamber cathode (main cathode). It is a thermoelectric source to start and maintain the electric rocket's discharge for steady operation. The other type is the neutralizer's hollow cathode. Its function is to inject an equally charged electron flow into the ion beam sprayed out from the electric rocket so as to avoid aircraft electrification.

The common requirements for the two types of cathodes are:

(1) The transmission density is high and the transmitted body is in the same maceration assuming the surface area has uniform transmission (actually it is very non-uniform).

(2) Long life-time and high reliability

(3) The cathode heater can withstand repeated electric current impact and the cathode can withstand repeated strong heat and cold impact. In a cold state, the cathode heater's resistance value is only $\frac{1}{5}$ - $\frac{1}{7}$ the resistance value when there is working temperature. When the cathode begins to heat, the impact current is large which causes the heater to vibrate. This in turn causes the heater to break, the welded points to come off and the aluminum oxide layer to peel off.

The cathode is located outside the aircraft casing and its lowest temperature can reach -30°C (keeps the mercury from

condensing). Within 5 to 10 minutes, the temperature rises to 1,000-1,200°C and in long-life the cathode need to withstand the repetition of this type of repeated, strong cold and heat impact. This can easily bring about changes in the material's crystal phase and the material becomes brittle and distorted. Therefore, the cathode has very stringent technical and structural requirements.

(4) The cathode endures ion bombardment, especially the neutralizer cathode. Its ring electrode voltage is twice that of the discharge chamber's ring electrode voltage. During the later period of the cathode's life after performance declines, its ring electrode voltage and (? word missing) voltage both rise. This causes a further increase of the ion energy of the bombarded neutralizer cathode. The ion bombardment can not only cause the cathode performance to decline but can also cause the cathode's welded seams to break. The size of the small holes on the top of the cathode expand or shrink and distort causing the parameters of the cathode to change.

(5) It can withstand the impact and vibration when an aircraft is launched. This impact and vibration can cause the oxide coating to possibly obstruct the small holes on the top of the cathode because of the launched body's fallen powder. This can cause the cathode to be non-operative.

(6) Stellar energy sources are limited. Their power consumption requirement is low, that is, the energy-launch ratio W/A must be small.

(7) Before an aircraft is launched, the cathode can exist in the atmosphere for a certain time without degenerating.

The manufacture of the cathode is of crucial importance in

the testing of the cathode. Several tens of cathodes were tested in (? illegible) and mercury steam and there were no heater and tantalum tube short circuits or outer layer aluminum oxide peeling. The reader should refer to our essay on the testing of cathodes.

(4) Research on the Distribution Regulating Separate System

This is an important separate system of the ion propulsion system and is the key to whether or not the electric rocket engine can be used in space. We must first study suitable ground test power sources to guarantee the success of the electric rocket engine ground tests, and on the basis of a large number of ground tests, develop research work on the use of the distribution regulating separate system in space.

(5) Comprehensive Tests of the System

In the final stage in the development of electric rocket components, these components must be joined together and act as a system in order to study the problem of determining the compatability and the interaction between the separate systems. Naturally, a small number of design changes can be permitted during testing.

6. Testing Long-Life and Reliability

In order for the electric rocket to be able to fulfill the operational long-life required by a space flight mission and convince people of its reliability, aside from using a series of long time endurance tests, clarify the mechanism of losing effectiveness in each component, and making structural, material and technical improvements, it is also necessary to research and evaluate long-life characteristics as well as use mathematical

statistical methods to carry out quantitative estimates of the reliability. For this reason, it is necessary to build test equipment to fulfill the requirements of long-life testing. The requirements for this equipment are:

(1) The corrosion of the sputtering objects should be as small as possible.

(2) Because of the influence of the residual gas on the electric rocket and long-life, the test vessel should have the environmental vacuity required for the tests.

(3) As far as possible reduce the test vessel's proportion of harmful gas component for the electric rocket.

(4) Raise the test equipment's operating reliability and the monitoring control capabilities required for the long-life tests as well as reduce equipment breakdowns.

Long-life test equipment have the following several major requirements:

(1) The test vessel (including the main cabin, delivery cabin, distribution cabin, mercury target, cold wall, shut-off valve and other equipment).

(2) Vacuum pumping equipment.
(3) Liquid nitrogen supply system
(4) Heating mercury dispersion system
(5) Monitoring control system
(6) Other auxiliary equipment

We should also consider multiple usage in the long-life tests, that is, should also be able to carry out dynamic state heat vacuum environment tests on the electric rocket.

Aside from this, when considering the situation wherein the electric rocket is mounted in the aircraft and is launched from the ground, it must also withstand impact and vibration tests. The development of test work for the electric rocket mechanical environment is also necessary.

(2) Conceptions and Views

In order to be able to further develop research on the electric rocket (ion engine), we feel that it is necessary to undertake work in the following areas:

(1) Develop Basic Scientific Research on the Electric Rocket

1. Research on Low Density Discharge Plasma

Low density discharge plasma physics is the physical basis of the electric rocket's ion source. It deals with the electric rocket's electric efficiency, the propellant's utilization rate and related performances.

2. Research on Magnetofluid Mechanics

To develop and lay a foundation for the research of the magnetic plasma electric-rocket.

3. Ion Optics Research

4. Research on High Vacuum Insulation

The third item of research is closely connected to the fourth item of research.

(2) Development of Research on the Utilization of and Technology Related to the Electric Rocket

1. Mission Analysis

Starting from the actual situation of China's national economy, formulate realistic plans for the utilization of and research on the electric rocket.

2. Establish advanced test measures required for electric rocket development using microcomputers as the basis.

3. Research a distribution regulating system with high reliability, high efficiency, light weight and which is fully automated.

4. Strengthen research on electric rocket manufacturing technology.

5. Research on the application of ion beams

By expanding the range of the ion beam energy and the beam current, it can be used in many industries. For example, ion beam etching, ion surface crystallization, ion beam polishing, ion beam welding.

6. Resolve the problem of providing an effective measure for the active control of the synchronous satellite charge. For example, to providean electric rocket or neutralizer.

(3) Other

1. Understand Major Foreign Developments in Electric Propulsion

Nuclear-electric propulsion is the final system operating in the solar system. It's constant power is unrelated to solar strength, it has long life, it does not endure radiation damage

and it does not require an added energy storage system or solar tracking system. It is suitable for orbit transport systems, earth orbit platforms, lunar and stellar bases, and stellar orbit research stations. It is necessary to lay a foundation in order to develop research on the use of the magnetic plasma electric rocket for primary propulsion.

2. Understand foreign advances in earth synchronous satellite solar power stations and where the special requirements for electric rockets are being placed.

3.When conditionally allowable, actively participate in international cooperation and research on electric rockets. Study from foreign workers researching electric rockets so as to supplement deficiencies in our own work.

The above is the work and conceptions we have accomplished in the field of electric rockets (ion engines). It is very incomplete. We invite the criticism and advice on our proposals from Japanese workers researching electric rockets.

Upon completing my report, I would like to wish the best for your country's greater success in ion engine research and success in your country's ETS-III satellite flight tests carrying ion engines. May the friendship between the people and scientific workers of China and Japan be everlasting!

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E403 AFSC/INA	1
E404 AEDC/DOF	1
E408 AFWL	1
E410 AD/IND	1
E429 SD/IND	1
P005 DOE/ISA/DDI	1
P050 CIA/OCR/ADD/SD	2
AFIT/LDE	1
FTD	
CCN	1
NIA/PHS	1
NIIS	2
LLNL/Code L-389	1
NASA/NST-44	1
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电火箭（离子发动机）的研究

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一、概况

电火箭又名电推力推进或说电推进（有的国家叫离子发动机），它是一种高比冲、低推力、长寿命的火箭发动机。

电火箭发动机推力的建立与任何其它形式的推进发动机（液体火箭发动机，空气推进发动机，核推进发动机等等）一样是依靠工质的加速和喷射，工质的加速和喷射或者用电加热的方法，或者用电和磁的体积力的方法。因此，根据电火箭的定义可分为三种基本类型，即

1、电热火箭，推进气体被电加热，然后在合适的喷嘴处膨胀产生推力。

2、静电火箭（离子发动机），由外加静电场加速喷射带电的推进剂粒子（离子）产生推力。

3、电磁火箭，由外磁场和内磁场与电流的相互作用推动电离了的推进剂流使之加速产生推力。

电火箭发动机的共同特点是：

1、电火箭发动机属于能源与工质分开的发动机。因此，它们的经济性不像热化学推进发动机（液体推进发动机，固体推进发动机和空气推进发动机）那样，只取决于一个参数——比冲，而是取决于两个独立的参数——比冲和效率（即把输入的电功率转换成推力的效率）。

比冲 $$I_s = \frac{T}{\dot{M}_p g_0} \qquad (1)$$

效率 $$\eta_e = \frac{P_B}{P_t} = \frac{\frac{1}{2}\dot{M}_p v_B^2}{P_t} = \frac{\text{流束的有效动能}}{\text{输入的电功率}} \qquad (2)$$

2、在任何条件下都希望有尽可能大的效率值。同时在一定条件下比冲具有最佳值（对应于能源系统的最小质量），这个最佳值与某些能源装置和发动机装置的原始数据有关，也与电火箭发动机推力作用下确定空间飞行器于定机动参数有关。

3、要得到比液体火箭发动机大很多倍的电火箭发动机的比冲原则上是可能的，因为供给工质的能量高度的集中。理论上，工质能量的集中和喷射的速度是不受限制的（因为电能无损失的转换为机械能—动能理论上是可能的）。

实际上，在电火箭发动机中，自然在加热某个结构部件中（在静电火箭发动机中）要发生焦耳损失和流动损失。在电磁火箭发动机中焦耳损失主要部分出现在导电的工质（等离子体）上。因此在电火箭发动机中（主要在电磁火箭发动机）中有一个供给能量的最高限，这是由于受结构材料耐热性的限制。但这个极限对应的工质流动速度比热推进发动机大很多倍。所以电磁火箭发动机的比冲能达到约10^4秒，而静电火箭发动机的比冲可达10^5秒（损失小，所以比冲高）。

4、电火箭发动机的特点有很大的比质量 。

$$\alpha = \frac{M_W}{P_t} = \frac{2\eta_e M_W}{T v_B} \qquad (3)$$

电火箭发动机的推力很小，提供能源的动力装置的质量很大，趋

过推力很多倍，因此电火箭只能用于制动，有效负荷大的场合，只能在空间使用。

针对电火箭的分类和它们的共同特点，我们在调查和总结国外发展电火箭的基础上，根据我们的具体情况提出研制电型电火箭（即离子发动机），这是基于离子发动机有一系列原理上的优点，它们主要是：在高的效率值下能保证大的比冲值（2000～10^5秒，或者说在这一比冲范围内有较高的效率）；工作过程的温度相当低，实际上消除了结构元件的冷却问题；能多次重复启动（国际上已接近10000次），连续地或间断地工作；寿命长（超过10000小时）；推力小，需要的推进时间长，作飞行器的控制时，可提高控制精度。我们研究离子发动机的主要目标之一是它的空间应用，并做一些基础研究。

二、电火箭（离子发动机）研究中的一些特点

(一)任务分析

随着轨道、飞行器和推进系统设计精度的不断增长，必须进行任务分析，以确定电火箭的使用场合和研制途径。任务分析在某种意义上来说类似于由许多有关的分系统组成的复杂的完整的系统设计。完成任务分析首先要解决任务的现实性和最后详细地设计飞行任务。

任务分析典型地从轨道到所希望的目标的参数研究开始。从这种研究中找到电源大小，飞行器重量，需要的助推飞行器种类，飞行时间和比冲的一般范围，而后飞行器和推进系统的初步文件设计将显示它是否

能够用现有的技术完成具有这些参数的任务。如果能行的话，就完成飞行器和推进系统的详细设计，以便在有关的无关的变量之间，例如重量、推力器的型号和数量、太阳阵设计、电源大小等与比冲、发射数据、飞行时间等之间确定现实的折衷。然后，这种设计研究成为更细心的轨道分析的基础，来用适合于飞行器设计的真正的推力器数目。接着从这种轨道分析将得到一组改进的任务分析曲线，由它完成第二次详细的飞行器设计。通常用第二次的飞行器设计完成轨道的最后决定（如果需要的话再重复一次）。

任务确定之后就对电火箭分系统的参数进行了解以便得到完整的推进系统的分析模型。最终的最佳设计必须同时做到使推进系统和轨道最佳化。

(二)设计计算

根据任务分析所确定的对象，首先估算一下电火箭发动机所需要的推力T，可用公式：

$$T = 0.65\times10^{-2}\,\frac{m_s}{\Delta\tau}\,g \quad \text{（半连续的推进修正）}$$

$$T = 3.59\times10^{-9}\,m_s\,g \quad \text{（连续的推进修正）} \qquad (4)$$

$$T = 1.63\times10^{-4}\,\frac{m_s}{\Delta\tau}\,g \quad \text{（脉冲的推进修正）}$$

接着对电火箭发动机的参数进行一系列的计算：

1、推力

$$T = \frac{d}{d\tau}(M_p V_B) = \dot{M}_p^{+} V_B \qquad \text{（牛顿）} \qquad (6)$$

2、比冲

$$I_s = \frac{T}{\dot{M}_p\, g}$$

（秒）　(7)

3、推进剂利用率

$$\eta_m = \frac{\dot{M}_p^+}{\dot{M}_p} \qquad (\%) \qquad (8)$$

4、束流

$$I_B = e\frac{\dot{M}_p^+}{M_p} = \frac{e\,\eta_m T}{M_p\, g\, I_S} \qquad (安) \qquad (9)$$

5、束功率（输出功率）

$$P_B = I_B V_B = \frac{1}{2}\dot{M}_p^+ v_B^2 \qquad (瓦) \qquad (10)$$

6、束速度

$$v_B = \sqrt{\frac{2e}{M_p} V_B} \qquad (米/秒) \qquad (11)$$

7、推力器电源效率（可见(2)式）

$$\eta_e = \frac{I_B V_B}{P_t} \qquad (\%) \qquad (12)$$

8、放电功率

$$P_D = I_D V_D \qquad (瓦) \qquad (13)$$

9、离子产生能量

$$\varepsilon_B = \frac{I_D V_D}{I_B} \text{ 或 } = \frac{I_D V_D + I_{DK} V_{DK}}{I_B} \qquad (eV/ion) \qquad (14)$$

10、推力器总效率

$$\eta_t = \eta_e \cdot \eta_m$$

$$\eta_t = \eta_e \cdot \eta_m \cdot \gamma^2 \qquad (\%) \qquad (15)$$

11、推进剂流量（等效安培表示）

$$= 9.649\times10^7 \times \frac{\dot{M}_p}{\mu} \qquad (安)$$

$$= 1.547\times10^{[illegible]} \frac{G}{T} \qquad (安) \qquad (16)$$

12、总的输入功率

$$P_t = I_B V_B + I_{ac} V_{ac} + (I_B + I_D) V_D + I_{DK} V_{DK} + I_{DH} V_{DH} + I_{NV} V_{NV} + I_{NK} V_{NK} + I_{NH} V_{NH} + I_{NV} V_{NV} + 其它 \qquad (瓦) \qquad (17)$$

关于电火箭的设计计算还有其它许多公式，这里就不一一列举。

通过上述计算就可以考虑电火箭样机（包括性能样机、工程样机、飞行样机三种类型），按照任务分析的要求，每一种样机都必须进行严格的地面试验，使电火箭日趋完善，最后达到空间使用要求。

我们选择阳极直径为8厘米的离子发动机，推力在3～6mN之间。

㈢电火箭（离子发动机）各分系统和工作内容

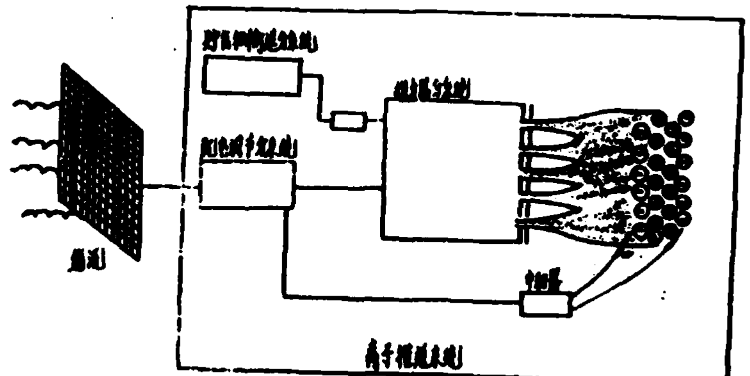


图1. [illegible]

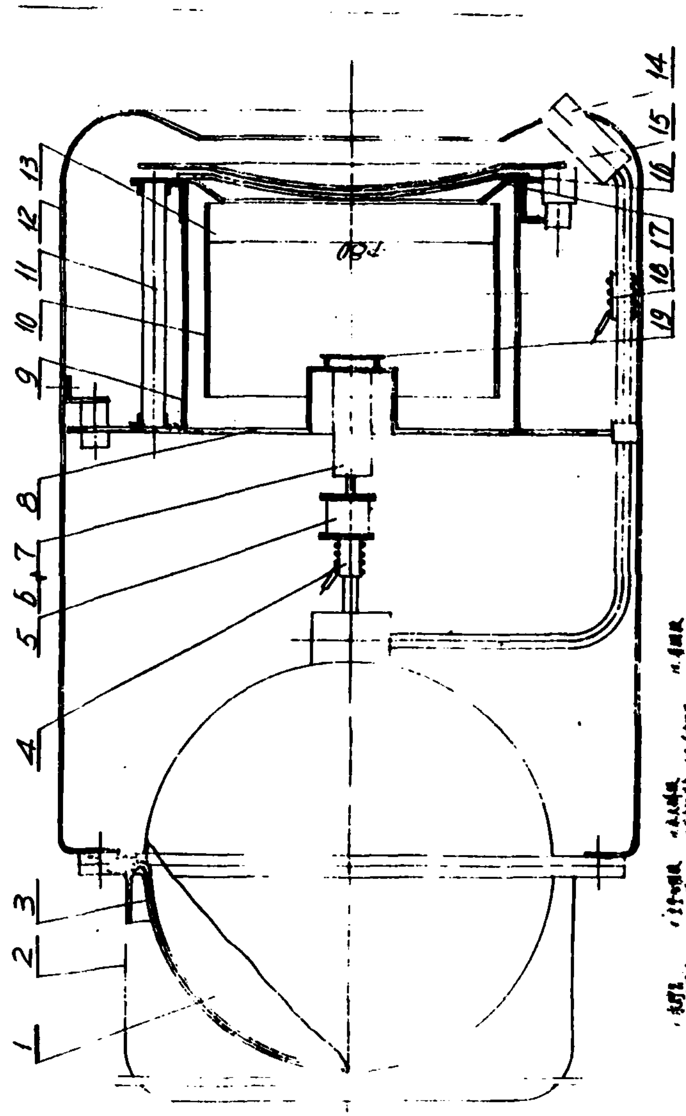
电火箭的基本组成为能源和离子推进系统（见图1）。能源应由专门的单位进行此项工作。离子推进系统是离子发动机所要研究的主要内容。它分为三个主要的分系统：(1)推力器分系统；(2)贮罐和输送分系统；(3)配电调节分系统（对推力器分系统、贮罐和输送分系统提供电源和控制）。

推力器分系又包括离子源，离子加速系统（或叫离子引出系统）和中和器。图2指出除配电调节分系统外的电火箭（离子发动机）结构示意图。我们认为电火箭（离子发动机）的工作内容有下列几个方面：

1、离子源研究和实验

离子源主要由阴极、阳极、放电室、极靴、磁钢和推进剂分配器所组成。阳极相对于两端板保持正电位使得阴极发射的电子通过阴极前面存在的等离子体鞘层进入电位随半径和轴向距离变化的区域。采用近似的轴向磁场防止电子直接达到圆筒状阳极。在这种交叉的电磁场结构中电子将螺旋形地围着磁力线并作纵向地漂移。由于电子和推进剂气体原子之间的电离碰撞的结果，将连续地提供有着离子和电子的等离子体，初始电子将损失能量和穿过磁力线扩散到阳极上并被收集，引起从阳极到阴极的电流。这种中性的等离子体充满了放电室。原则上，这种放电装置能够被用来使任何气体电离；实际上，最经常地采用汞和水银作为推进剂，我们是用水银作推进剂。

离子源的研究包括：离子源放电室物理模型的探讨；放电室的离



子溅射机制的探讨；双荷离子成分的理论计算（[redacted]）；离子引出系统中离子光学的计算机模拟（研究束发散对推力的影响）；加速极下游表面电荷交换离子轰击溅射的理论计算；离子束中和机制的了解；高真空绝缘机制的了解。

离子源的实验包括：离子源的性能试验和性能最佳化试验，试验的目的是设计出最佳化的离子发动机结构，以使与配电调节分系统一起作离子发动机的联合试验和整机热性能试验。离子源实验是离子发动机研究工作中的主要工作内容，不论國外，还是我们都把工作的重点放入到实验中去。

有了最佳化的离子源，还必须有好的离子引出系统。通常离子引出系统是根据束电流、束电压、比冲、阻止电子逆流所需的最小加速电压（$V_{ac\,min}$），引出束流所需的最小总加速电压（$V_{t\,min}$），屏栅开孔面积比（F_{os}），最大净——总加速电压比（V_n/V_t），束发散角（α），影响放电的因素（推进剂利用效率、放电损耗、放电电压）、经受的发射环境和反复的热循环等因素综合考虑的。根据下列方程式（从查尔特定律推导出来的方程式）来描述离子引出系统的特性：

电流密度：

$$J_B=\frac{4\varepsilon_0}{9}\left(\frac{2q}{m}\right)^{1/2}\cdot\frac{V_t^{3/2}}{l_e^2} \qquad (18)$$

电流（单孔的电流）

$$I_B=\frac{8\varepsilon_0}{9}\left(\frac{q}{2m}\right)^{1/2}\cdot\frac{A_b}{l_e^2}\cdot V_t^{3/2} \qquad (19)$$

$$I_B=\frac{\pi\varepsilon_0}{9}\left(\frac{2q}{m}\right)^{1/2}\cdot\frac{d_s^2}{l_e^2}\cdot V_t^{3/2}$$

考虑到离子源工作时的热变形，离子引出系统的两个栅极（屏栅和加速栅）均做成盘状，为了降低制成盘状栅极所引起的推力损失，要求对盘状栅极进行变形补偿，补偿值为：

$$\delta_1=\frac{2r_H h^2\left(\frac{2g}{r_b^2+h^2}+\frac{t_s+t_{ac}}{r_b^2-h^2}\right)}{(r_b^2+h^2)\arcsin\left(\frac{2r_b h}{r_b^2+h^2}\right)} \qquad (20)$$

$$\delta_2=\delta_1+\frac{2hd_{ac}}{(r_b^2+h^2)K}$$

我们的离子光学系统的计算是在Z—80微型计算机上进行的。

2、蒸发器研究和实验

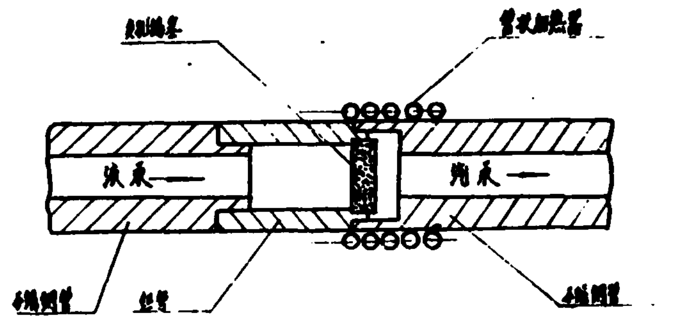


图3 蒸发器示意图

图3是蒸发器的结构示意图。它是离子发动机输送系统中的一个部件，它的作用是作为汞的液汽分离器，将来自贮箱的液态汞加热蒸发为推力器放电室及空心阴极提供一定流率的汽态汞，它还可以阻止液汞进入放电室。

蒸发器的试验工作包括：

(1)空气流量试验，测量不同压力下空气流过蒸发器的速率。

(2)汞流量试验，蒸发器汞流量是温度的函数，其流量大小是由加热器控制的。

(3)渗透试验，试验液汞渗透多孔钨的极限压力（汞与钨是不浸润的，故液汞接触到多孔钨毛细管时不能通过，直到液汞压力超过液汞弯月面形成的附加压力为止）。渗透压力可达1 0个大气压。

(4)渗透试验后重做汞流量试验，以证明多孔钨渗透前后汞流量基本上没有变化。

(5)寿命试验，主要是试验结构的合理性，长期工作后流通特性的变化，蒸发器的寿命究竟多长？

我们的蒸发器寿命已超过6 5 0 0小时，蒸发器的汽汞流量基本上没有变化，能满足电火箭试验时的要求。

3、阴极研究

根据任务分析和电火箭的工作特点，要制作两种阴极，一种是放电室阴极（主阴极），它主要是一个热电子源，启动并维持电火箭放电，使之稳定地工作；另一种是中和器空心阴极，它的作用是向电火箭喷出的离子束中注入等量的电荷电子流，避免飞行器带电。

对两种阴极的共同要求是：

(1)发射密度高，同浸渍内发射体，假定表面积均匀发射（实际很不均匀）。

(2)、寿命长，可靠性高。

(3)阴极加热器能经受多次电流冲击及阴极能经受多次剧烈的冷热冲击。冷态时，阴极加热器电阻值只有工作温度时阻值的$\frac{1}{5}$——$\frac{1}{7}$，阴极开始加热时，冲击电流大，使加热器振动，引起加热器断裂，焊点脱落，氧化铝层龟裂、剥落。

阴极处于飞行器壳体外，其最低温度可达－3 0℃（保持汞不冷凝）。在五到十分钟内，升温至1000～1200℃，阴极在其寿命期，要经受多次这样反复冷热的剧烈冲击，容易引起材料晶相改变，使材料变脆，变形，也给阴极工艺、结构上带来严格的要求。

(4)阴极耐离子轰击。尤其是中和器阴极，其环电极电压较放电室环电极电压高一倍，在阴极寿命后期，性能衰减后，其环电极电压和合电压都将上升，这样将进一步引起轰击中和器阴极的离子能量加大。离子轰击不但会使阴极性能下降，还可能使阴极焊缝遭到破坏，阴极顶部小孔尺寸扩大或缩小，畸变，导致阴极参数改变。

(5)能经受飞行器发射时的冲击和振动。冲击和振动会造成氧化物涂层内发射体因掉粉可能堵塞阴极顶小孔，造成阴极不工作。

(6)星上能源有限，功耗要求低，即功率发射比W／A要小。

(7)飞行器发射前，阴极能在大气中保存一定时间，不变质。

在阴极的实验过程中，阴极的制作工艺是个关键。在氩气和汞蒸汽中试验了数十只阴极，没有出现加热器与钽管短路，外层氧化铝剥落等现象。有关阴极的实验请参考我们带来的文章。

4、配电调节分系统的研究

它是离子推进系统的一个重要的分系统，是电火箭发动机在空间能否成功使用的关键。首先必须研究合适的地面试验电源以保电火箭发动机地面试验的顺利进行，在大量地面试验的基础上，开展空间用配电调节分系统的研究工作。

5、系统组合试验

在电火箭部件研制的最后阶段，这些部件必须连在一起和作为一个系统来研究以便决定相容性和分系统的相互作用问题，当然试验过程中仍旧可以允许少量的改变设计。

6、寿命和可靠性试验

为了使电火箭能满足空间飞行任务要求的运转寿命，并达到令人信服的可靠性，除了通过一系列长时间耐久试验，弄清各部件的失效模式与失效机理，在结构、材料、工艺方面作大幅改进之外，还要进行寿命特征的研究并定以及采用数理统计方法进行可靠性的定量估算工作。为此，必须建立满足寿命试验要求的试验设备。对该设备的要求是：

(1)溅射物对电火箭的腐蚀应尽可能的小。

(2)由于剩余气体对电火箭性能与寿命有影响，所以试验容器应具有试验所要求的环境真空度。

(3)尽可能减少试验容器中对电火箭有害的气体成分的比例。

(4)提高试验设备运转可靠性以及寿命试验所要求的监测控制能力，减少设备故障。

寿命试验设备包括以下几个主要部分：

(1)试验容器（包括主舱、付舱、配电舱、水银舱、冷壁和供电导线等）。

(2)真空抽气系统。

(3)液氮供给系统。

(4)加热驱汞系统。

(5)监测控制系统。

(6)其它辅助设施。

寿命试验设备还应考虑到多用途，即还能进行电火箭的动态热真空循环试验。

此外，考虑到电火箭装在飞行器上，由地面发射时还要经受冲击，振动等因素的考验，开展电火箭力学环境的试验工作是必须的。

三、设想与看法

为使电火箭（离子发动机）研究工作能深入地开展下去，我们认为应该进行下列几方面的工作：

(一)开展与电火箭有关的基础科学的研究

1、低密度放电等离子体的研究

低密度放电等离子体物理是电火箭离子源的物理基础，它涉及电火箭的电效率，推进剂利用率和有关的性能。

2、磁流体力学的研究

为开展磁等离子体电火箭研究奠定基础。

3、离子光学研究

4、高真空绝缘的研究

第3项研究与第4项研究密切相关。

(二)开展与电火箭有关的应用和技术研究

1、任务分析

从我国国民经济的实际情况出发，制订出切实可行的电火箭应用和研究计划。

2、建立以微型计算机为基础的符合电火箭研制要求的先进的实验手段。

3、高可靠性、高效率、全自动化、轻重量的配电调节分系统的研究。

4、加强电火箭制作工艺的研究

5、离子束应用研究

由于束能量和束电流的广阔范围，使得它能够应用到许多工业上。如离子束刻蚀、离子表面结晶、离子束抛光、离子束焊接。

6、为解决同步卫星带电的主动控制技术提供有效的实施手段。譬如提供电火箭，或者中和器。

(三)其它

1、了解国外主电推进的进展情况

核电推进是太阳系中运输的最终系统。它有着恒定的功率与太阳强度无关，寿命长，不会受到辐射损坏，不需要辅助的能量贮存系统或精

确的太阳跟踪系统。适合于轨道运输系统，地球轨道平台，月球和行星基地以及行星轨道研究站。为开展主推进用的磁等离子体电火箭的研究打下基础。

2、了解国外关于地球同步卫星太阳电站研究的进展情况。预见将对电火箭提出特殊的要求。

3、在条件许可的情况下积极参与国际上电火箭的合作和研究。向国外研究电火箭的同行们学习，以弥补我们工作中的不足。

以上是我们在电火箭（离子发动机）研究方面所做的一些工作和想法，很不成熟，请日本研究电火箭的同行们对我们提出批评和指教。

我的报告完了，予祝贵国在离子发动机研究工作中取得更大的成功！予祝贵国带有离子发动机的ETS—Ⅲ卫星空间飞行试验成功！祝中日两国人民和科学工作人员之间的友谊万古长青！

END

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